END
DATE
FILMED
3-81
DTIC

CRACK PROPAGATION IN ELECTRON BEAM WELDED Ti-6Al-4V

UNDER SPECTRUM LOADING CONDITIONS

C. E. Neu
Aircraft and Crew Systems Technology Directorate
NAVAL AIR DEVELOPMENT CENTER
Warminster, Pennsylvania 18974

18 December 1980

PHASE REPORT
TASK NO. ZF54-590-001
Work Unit No. ZM601

AD A095585

81 2 27 040

NOTICES

REPORT NUMBERING SYSTEM - The numbering of technical project reports issued by the Naval Air Development Center is arranged for specific identification purposes. Each number consists of the Center acronym, the calendar year in which the number was assigned, the sequence number of the report within the specific calendar year, and the official 2-digit correspondence code of the Command Office or the Functional Directorate responsible for the report. For example: Report No. NADC-78015-20 indicates the fifteeth Center report for the year 1978, and prepared by the Systems Directorate. The numerical codes are as follows:

CODE	OFFICE OR DIRECTORATE
00	Commander, Naval Air Development Center
01	Technical Director, Naval Air Development Center
02	Comptroller
10	Directorate Command Projects
20	Systems Directorate
30	Sensors & Avionics Technology Directorate
40	Communication & Navigation Technology Directorate
50	Software Computer Directorate
60	Aircraft & Crew Systems Technology Directorate
70	Planning Assessment Resources
80	Engineering Support Group

PRODUCT ENDORSEMENT - The discussion or instructions concerning commercial products herein do not constitute an endorsement by the Government nor do they convey or imply the license or right to use such products.

APPROVED BY: _Edward Sturm_   DATE: _18 Dec. 80_
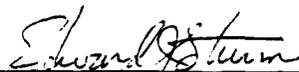
       E. J. STURM
       CAPT   USN

(12) 28

(14)

REPORT DOCUMENTATION PAGE		READ INSTRUCTIONS BEFORE COMPLETING FORM
1. REPORT NUMBER NADC-80202-60	2. GOVT ACCESSION NO. AD-A095585	3. RECIPIENT'S CATALOG NUMBER
4. TITLE (and Subtitle) CRACK PROPAGATION IN ELECTRON BEAM WELDED Ti-6Al-4V UNDER SPECTRUM LOADING CONDITIONS.		5. TYPE OF REPORT & PERIOD COVERED (9) PHASE REPORT.
		6. PERFORMING ORG. REPORT NUMBER
7. AUTHOR(s) CHARLES E. NEU		8. CONTRACT OR GRANT NUMBER(s)
9. PERFORMING ORGANIZATION NAME AND ADDRESS Aircraft and Crew Systems Technology Directorate Naval Air Development Center Warminster, Pennsylvania 18974		10. PROGRAM ELEMENT, PROJECT, TASK AREA & WORK UNIT NUMBERS AIRTASK NO. ZF54 590 001 Work Unit No. ZM601
11. CONTROLLING OFFICE NAME AND ADDRESS Naval Air Systems Command Department of the Navy Washington, DC 20361	(11)	12. REPORT DATE 18 December 1980
		13. NUMBER OF PAGES 24
14. MONITORING AGENCY NAME & ADDRESS(if different from Controlling Office) (16) F54590		15. SECURITY CLASS. (of this report) UNCLASSIFIED
		15a. DECLASSIFICATION/DOWNGRADING SCHEDULE

16. DISTRIBUTION STATEMENT (of this Report)

Approved for Public Release; Distribution Unlimited

17. DISTRIBUTION STATEMENT (of the abstract entered in Block 20, if different from Report)

18. SUPPLEMENTARY NOTES

19. KEY WORDS (Continue on reverse side if necessary and identify by block number)

Titanium   Crack propagation
Fatigue   Spectrum Loading
Electron beam welding   Ti-6Al-4V
F-14 Aircraft

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

Center cracked panels of electron beam welded Ti-6Al-4V were tested under
truncated F-14 aircraft wing fatigue spectra in an effort to simulate
propagation of fatigue cracks from "spike" flaws at weld intersect holes.
Predictions of fatigue lives of F-14 wing carrythrough sections for various
initial defect sizes were made using fracture mechanics methods.

393532
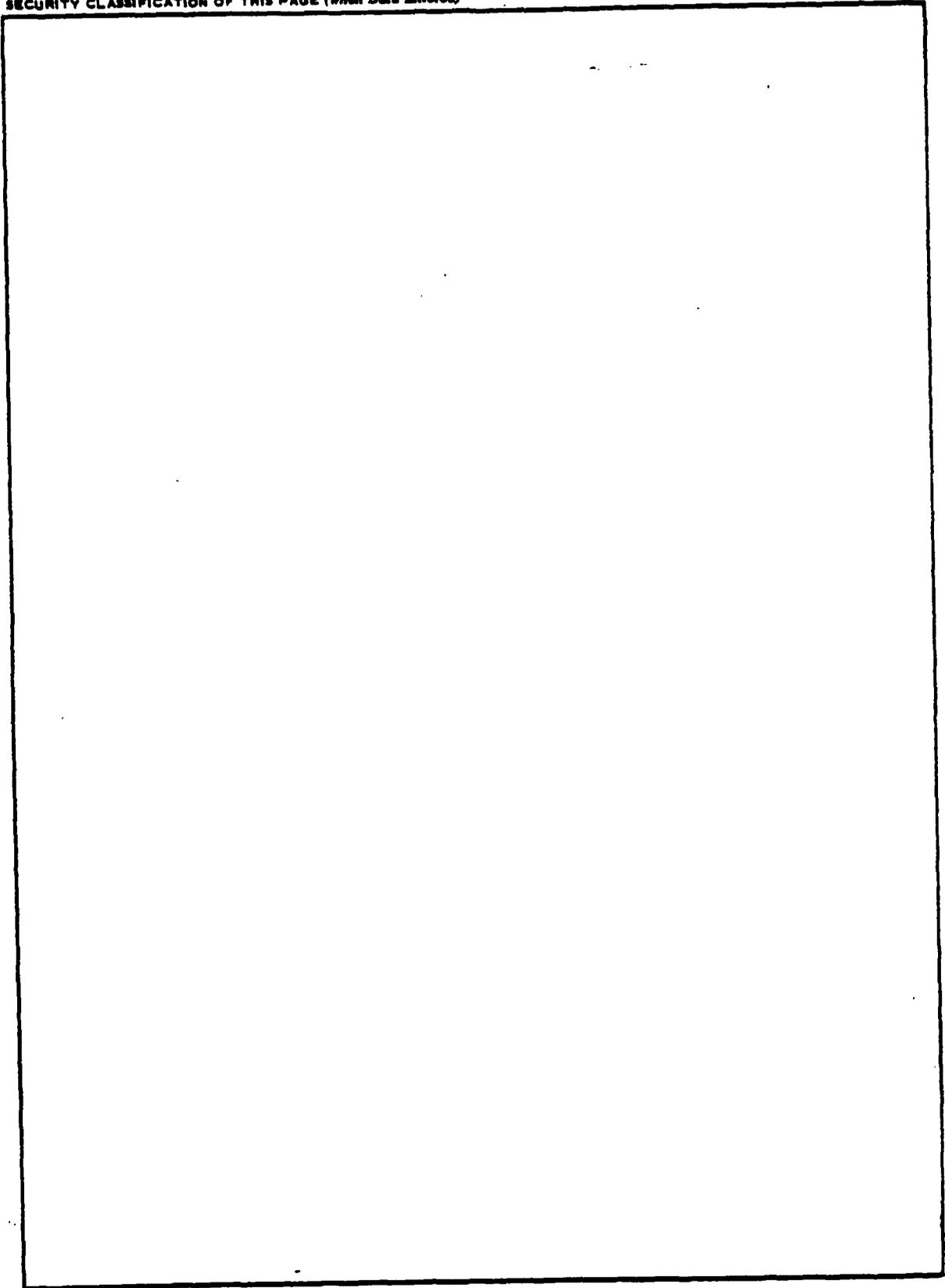
SUMMARY

INTRODUCTION

This test program was undertaken as a result of the premature failure of an
F-14A aircraft wing carrythrough section test article. The failure occurred by
fatigue crack propagation originating at a "spike" defect at a weld intersection
point. As shown schematically in Figure 1, a "spike" defect occurs unavoidably
when two electron beam welds intersect. After the first weld is completed, the
second weld is made with parameters set for thickness T1. When the electron
beam encounters the preexisting weld, it "sees" an effective thickness of T2,
the entire length of the first weld. A defect occurs at the limit of penetra-
tion of the beam. Standard procedure in fabrication of the wing carrythrough
section is to remove the defect with a drilled hole. In the case of the fatigue
test article, an incompletely removed defect at the surface of the hole escaped
detection and served as the failure origin.

It was thought that similar defects might be present in production aircraft. This
program was a study of the propagation rates of simulated electron beam weld
flaws in Ti-6Al-4V under F-14 wing spectrum loads in order to predict safe lives
of aircraft and to set revised inspection standards.

SUMMARY OF RESULTS

The results of the study using 4G and 5G loading spectra gave a total life
expectancy of 2928 - 5328 and 1370 - 2250 flight hours respectively. These data
included both initiation of cracks from machined notches and their propagation
to failure. The data spread indicates the scatter band for the experimental
results.

Residual life remaining after the first detection of fatigue cracking was
902 - 1528 and 50 - 810 flight hours respectively for the 4G and 5G spectrum
load tests.

Fracture mechanics prediction methods were used to predict fatigue life expectancy
of F-14 aircraft wing carrythrough sections (Table 6) for various size defects
at weld intersect holes.

CONCLUSIONS

1. Crack propagation controlled fatigue lives of structures under variable
amplitude loading conditions can be predicted from fracture mechanics specimen
tests, but the reliability of the method utilized herein is limited at present
by the requirement for successive graphical interpretations.

2. Weld intersect hole defects 0.354 inch (9.0 mm.) and larger for 1/4 inch
(6.35 mm) diameter holes, 0.079 inch (2.0 mm) and larger for 1/2 inch (12.7 mm)
diameter holes, and 0.039 inch (1.0 mm) and larger for 1.0 inch (25.4 mm) diameter
holes cannot be considered to have a safe life expectancy for even one Scheduled
Depot Level Maintenance (SDLM) interval for the F-14 aircraft.

TABLE OF CONTENTS

LIST OF FIGURES

LIST OF TABLES

EXPERIMENTAL PROCEDURES

A. MATERIAL

The base material for the test program was annealed Ti-6Al-4V in the form of two rolled plates each 1-3/4 x 12 x 52 inches (44.5 x 293 x 1346 mm). Chemical composition and tensile properties reported by the supplier are listed in Tables I and II. The plates were electron beam welded to form a 1-3/4 x 24 x 52 inch (44.5 x 586 x 1346 mm) panel by Grumman Aerospace Corporation (GAC) utilizing the same welding and inspection procedures used for thick-section welds on F-14 aircraft wing carrythrough sections. Inspection of the weld disclosed one burst type defect approximately 7/16 inch (11.1 mm) in diameter. The defect was drilled out so that all test specimens would be assured of being initially defect free.

TABLE I

CHEMICAL ANALYSIS (WEIGHT PERCENT) OF Ti-6Al-4V PLATE MATERIAL

Heat No.	Al	V	C	Fe	H_2	O_2	N_2	Ti
K-6795	6.2	4.2	0.026	0.20	0.002	0.17	0.009	Bal.

TABLE II

TENSILE PROPERTIES OF Ti-6Al-4V PLATE MATERIAL

Orientation	0.2% Offset Yield Strength ksi (MPa)	Tensile Strength ksi (MPa)	Elongation %
Longitudinal	137 (944)	145 (1000)	13
Transverse	130 (896)	139 (958)	12

B. SPECIMEN LAYOUT AND CONFIGURATION

Figure 2 shows the test specimen layout for the welded panel. The hole where the above mentioned burst defect was removed is indicated on the figure near the finish end of the weld.

The specimen configuration used (Figure 3) was a center notched panel with grip ends designed for clamping rather than the more common pin loading arrangement. Clamped ends were required in order to prevent rotation of the specimen under compressive loads and to avoid pin-slamming during transition between tension and compression.

The 1/2 inch (12.7 mm) center hole with crack starter notches was designed to simulate an incompletely removed "spike" defect at a weld intersection hole. The center portion of each specimen was swabbed with Keller's Reagent after surface machining was complete to locate the holes and notches on the weld centerline.

1

SPIKE DEFECT

WELD
INTERSECT
HOLE
(DRILLED AFTER
SECOND WELD
TO REMOVE
SPIKE DEFECT)

T_1

SECOND WELD

FIRST WELD

T_2
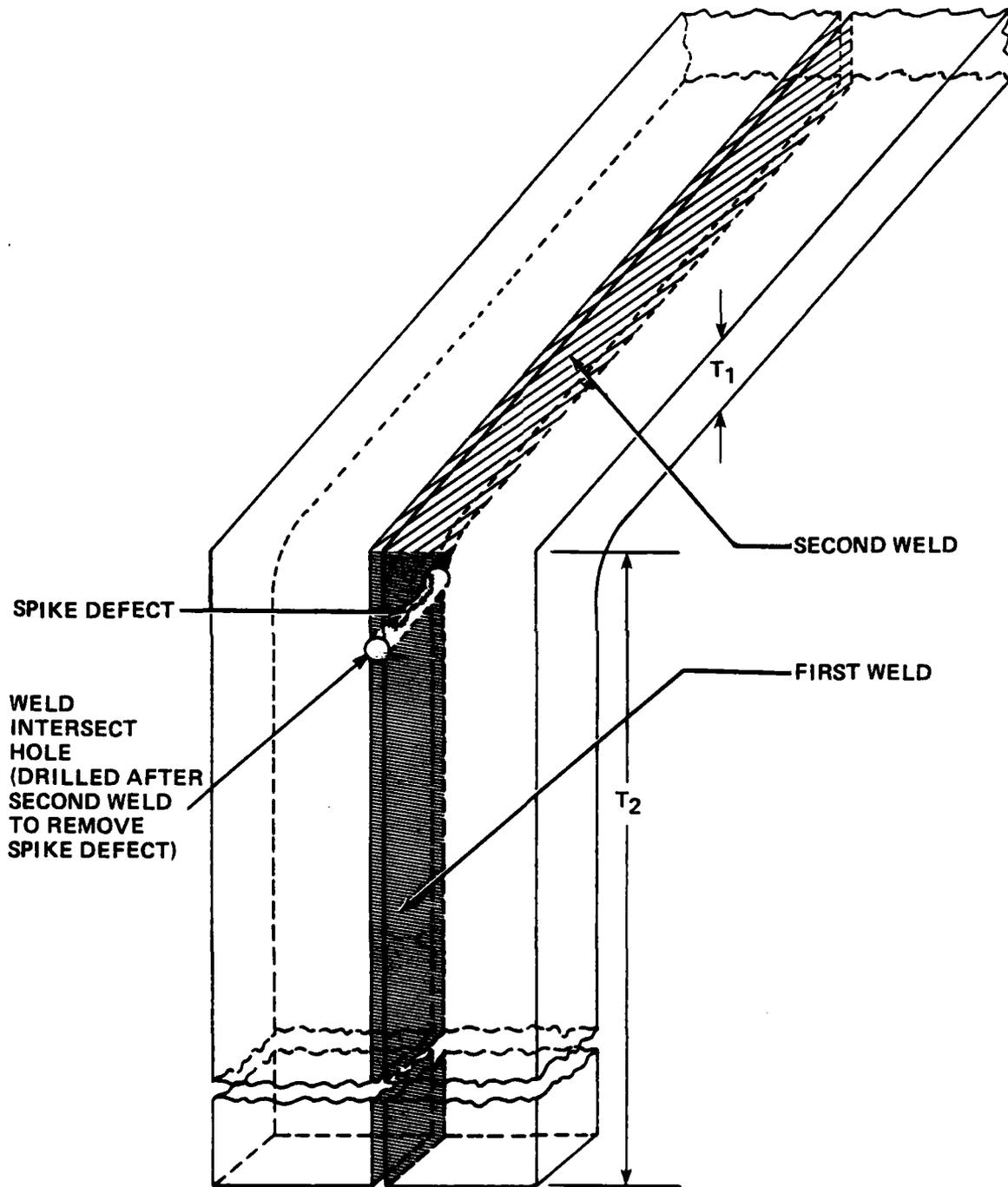
Figure 1. Spike Defects and Weld Intersect Holes

2

FW-8

FW-7

FW-6

FW-5

FW-4

FW-3

FW-2

FW-1

7/16" D HOLE

WELD

7"
(TO HOLE)

52"

24"
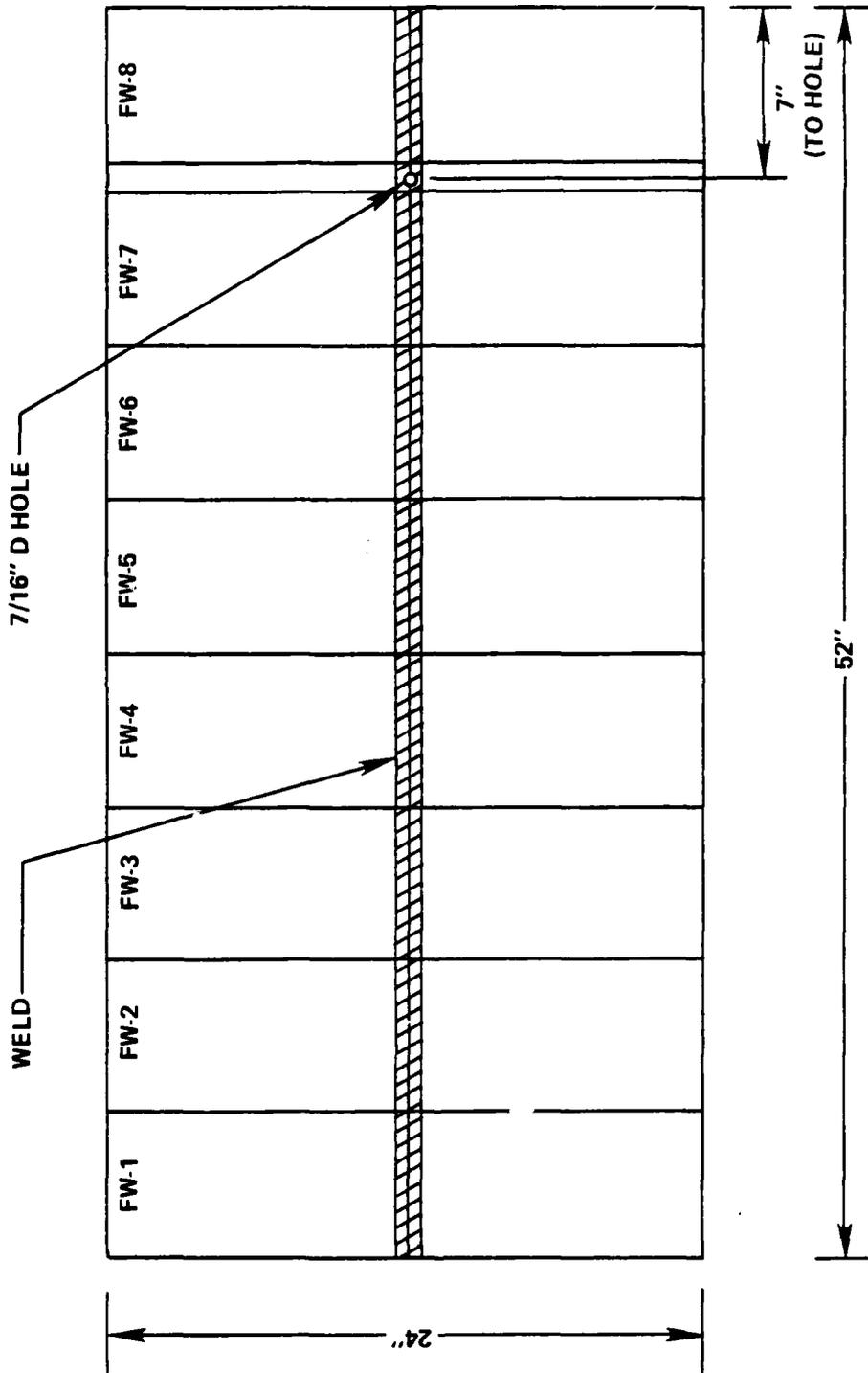
Figure 2. Titanium Electron Beam Weld Specimen Layout Diagram

NOTE: CRACK STARTER NOTCHES IN WELD ZONE MAY BE MACHINED V-NOTCHES OR JEWELER'S SAW CUTS.
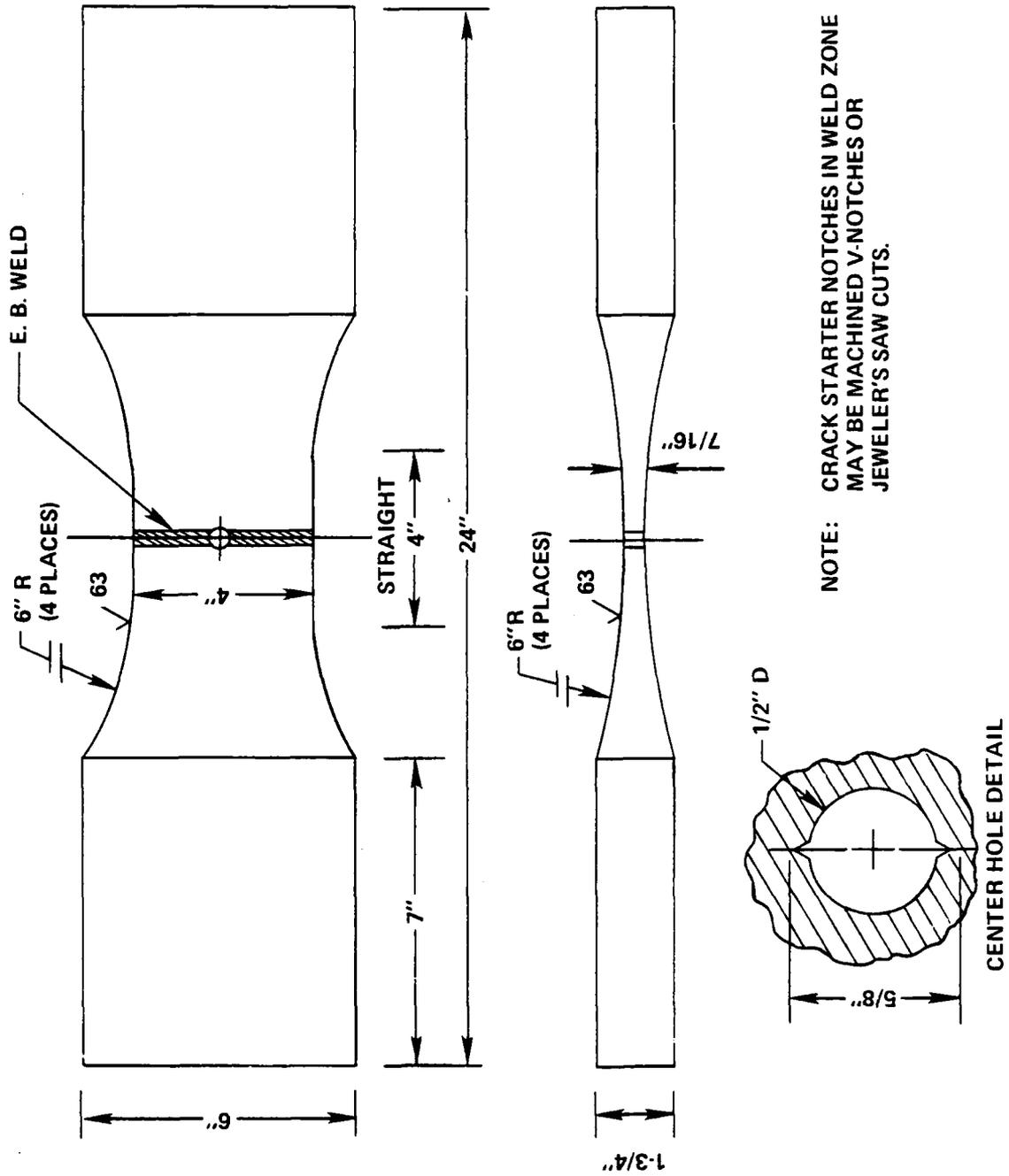
Figure 3. Titanium Flawed Electron Beam Weld Fatigue Specimen

4

C. TEST METHOD

1. Load Spectra

F-14 wing fatigue spectra truncated to 4G and 5G stress levels were pro-
vided by GAC (reference (a)) based on a limit stress of 38 ksi (262 MPa). Stress
levels together with number of occurrences per 200 flight hours are listed in
Tables III and IV. The sequence of application of the stresses was pseudo-
randomized* within a 200 flight hour block, and corresponding loads calculated
based on the gross area of the specimen (1.75 in.^{2}) (0.00113 m^2) were programmed
into a tape controlled EMR Instruments Model 1641 Profiler. The profiler pro-
vided command signal and both slope and end point error detection for a Budd Type UEH
closed loop servo-hydraulic test machine of 400,000 pound (1.79928 MN) capacity.
Loading was sinusoidal at a frequency of 1 to 2 hz.

TABLE III

WELD INTERSECT HOLE STRESS SPECTRUM TRUNCATION TO 4G

| Stress Level	Maximum Stress		Minimum Stress		No. of Occurrences 200 Flight Hr. Block
	psi	(kPa)	psi	(kPa)	
1	11,230	(77,428)	-370	(-2,551)	7
2	11,895	(82,013)	-6,670	(-45,988)	1
3	12,313	(84,895)	-4,817	(-33,212)	3
4	12,356	(85,192)	-2,964	(-20,436)	14
5	12,422	(85,647)	3,242	(22,353)	886
6	12,446	(85,812)	-1,204	(-8,301)	284
7	19,874	(137,026)	-6,080	(-41,920)	1
8	20,890	(144,032)	-3,040	(-20,960)	23
9	21,330	(147,065)	4,560	(31,440)	353
10	22,462	(154,870)	-2,128	(-14,672)	30
11	23,634	(162,951)	1,604	(11,059)	3
12	23,897	(164,764)	11,177	(77,063)	432
13	24,378	(168,080)	4,231	(29,172)	8
14	24,378	(168,080)	-2,128	(-14,672)	8

2. Flaw Propagation Measurements

As the specimens were cycled under spectrum loading conditions, observa-
tions were made with an optical micrometer to detect crack initiation at the
tips of the notches. Crack length measurements after initiation were made at
the end of each 200 flight hour block until the specimen failed. The surfaces
of the specimens in the area of observation were lightly grit blasted to reduce
glare.

* The application sequence was not entirely random in that the number of applica-
  tions of the lesser occurring loads was fixed.

5

TABLE IV

WELD INTERSECT HOLE STRESS SPECTRUM TRUNCATION TO 5G

Stress Level	Maximum Stress psi	(kPa)	Minimum Stress psi	(kPa)	No. of Occurrences 200 Flight Hrs. Block
1	11,230	(77,428)	-370	(-2,551)	7
2	11,895	(82,613)	-6,670	(-45,988)	1
3	12,813	(88,343)	-4,817	(-33,213)	3
4	12,986	(89,535)	-2,964	(-20,436)	14
5	13,166	(90,776)	-1,204	(-8,301)	284
6	13,222	(91,163)	3,242	(22,353)	886
7	19,874	(137,026)	-6,080	(-41,920)	1
8	21,850	(150,651)	-3,040	(-20,960)	21
9	22,480	(154,994)	4,560	(31,440)	323
10	23,442	(161,627)	-2,128	(-14,672)	37
11	23,634	(162,951)	1,604	(11,059)	3
12	24,227	(167,039)	11,177	(77,063)	432
13	29,032	(200,169)	4,560	(31,440)	38
14	29,032	(200,169	-2,668	(-18,395)	3

DATA ANALYSIS AND RESULTS

Three specimens, FW-3, FW-5 and FW-7, were tested under the 4G spectrum and four, FW-2, FW-4, FW-6 and FW-8, under the 5G spectrum. Specimen FW-1 was subjected to an accidental high load and broke before any crack propagation measurements could be made. The raw data from each test consisted of a series of recorded crack length measurements with corresponding elapsed blocks or flight hours; these data are shown in Table V.

TABLE V

FATIGUE CRACK LENGTH MEASUREMENTS FOR
Ti-6Al-4V ELECTRON BEAM WELD SPECIMENS

Specimen No.	Spectrum	Flight Hours	Total Crack Length (2 1) inch (mm)	
FW-3	4G	1800	0.6213	(15,781)[1]
		2000	0.6494	(16,496)
		2200	0.6418	(16,301)
		2400	0.6719	(17,066)
		2600	0.7115	(18,073)
		2800	0.7487	(19,017)

Specimen failed at 2928 flight hours;
Failure stress: 23,897 psi (164,764 kPa)

TABLE V (cont'd)

Specimen No.	Spectrum	Flight	Total Crack Length (2 1) inch	(mm)
FW-5	4G	3600	0.6380	(16,205)[1]
		3800	0.7267	(18,457)
		4000	0.7514	(19,086)
		4200	0.7815	(19,851)
		4400	0.8531	(21,669)

Specimen failed at 4502 flight hours;
Failure stress: 23,897 psi (164,764 kPa)

FW-7	4G	3600	0.6396	(16,246)[1]
		3800	0.6500	(16,509)[2]
		4000	0.6579	(16,710)[2]
		4200	0.6615	(16,802)[2]
		4400	0.6774	(17,207)[2]
		4600	0.6839	(17,371)[2]
		4800	0.7088	(18,004)
		5000	0.7453	(18,931)
		5200	0.8117	(20,616)

Specimen failed at 5328 flight hours;
Failure stress: 23,897 psi (164,764 kPa)

FW-2	5G	1000	0.6251	(15,877)[1]
		1200	0.6394	(16,240)
		1400	0.6449	(16,380)
		1600	0.6626	(16,830)
		1800	0.6784	(17,231)
		2000	0.7528	(19,120)

Specimen failed at 2020 flight hours;
Failure stress: unknown

FW-4	5G	2000	0.6373	(16,187)[1]
		2200	0.6551	(16,639)
		2250	0.7322	(18,599)

Specimen failed at 2250 flight hours;
Failure stress: 22,480 psi (154,994 kPa)

FW-6	5G	1200	0.650	(16.5)[3]

Specimen failed at 1370 flight hours;
Failure stress: 29,032 psi (200,169 kPa)

FW-8	5G	1000 + [4]	0.6798	(17,266)

Specimen failed at 1162 + (4) flight hours;
Failure stress: unknown

NOTES: (1) Starter notch length; no crack.
(2) Cracked only one side of starter notch.
(3) Starter notch length; measured after failure.
(4) No crack was observed after 1000 flight hours. Specimen was then subjected to 12,782 cycles between 286 psi (1972 kPa) minimum stress and 17,420 psi (120,107 kPa) maximum stress to initiate crack before 5G spectrum cycling was resumed.

In instances where the load at failure was known, final crack lengths were calculated based on an assumed fracture toughness (K_{Ic}) of 38 ksi $\sqrt{in}$. (41.8 MPa $\sqrt{m}$) for the electron beam weld. This assumption was based on earlier in-house testing of similar welds (reference (b)). No measurement could be made of the final crack length on the fractured specimens to verify the accuracy of the calculated crack lengths. As one can see in Figure 4, the similarity in appearance between the spectrum fatigue crack area and the fast fracture area made it difficult to accurately identify a line of demarkation by optical means.

Several analytical operations were performed to convert the data into a form useful for predicting weld defect fatigue behavior in F-14 aircraft. The elapsed flight hours corresponding to each crack length measurement were converted to flight hours remaining before failure, or life expectancies. The total crack lengths (2 1) were then plotted against the logarithms of the life expectancies so that the range of data points could be reasonably bounded by straight lines. The crack length vs. life expectancy curves are hyperbola-like and not amenable to graphical interpretation with linear coordinates. These plots are shown in Figures 5 and 6. From bounds of the data, ranges of life expectancies for any crack length could be inferred.

It was then necessary to provide a conversion from the geometry of the test specimen to that of a weld intersect hole in the wing carrythrough section. This was accomplished by correlating stress intensity factors (K_I). A stress intensity factor solution by Newman and Tada (reference (c)) was employed for the test specimen. This solution is designed for a finite width panel with a symmetrically cracked center hole under uniform tension and is illustrated in Figure 7. In the figure, the curve for c/b = .125, the hole radius to panel half-width ratio for the specimen, was constructed by interpolation.

The Bowie solution for a singly cracked hole (reference (d)), specified for uniaxial stress, was employed to simulate a hole in a weld intersection with an incompletely removed "spike" defect. This solution is illustrated in Figure 8. "Normalized" stress intensity factors were calculated for a specimen geometry over a range of total crack lengths (2 1) and for 1/4 inch (6.35 mm), 1/2 inch (12.7 mm) and 1 inch (25.4 mm) diameter weld intersection holes over a range of single crack lengths (a). For convenience unity was used for the stress term (σ) in all calculations, and all dimensions including crack lengths were in millimeters. Thus, the numerical results obtained for stress intensity factors (K_I) were comparable to each other, but had no relationship to actual stress intensity factors encountered by the specimens or the airplanes. Results of these calculations are plotted in Figures 9 and 10.

From Figures 9 and 10 it was then possible to find for any value of the specimen total crack length (2 1) the corresponding weld intersect hole defect size (a) that would result in the same stress intensity level (K_I) for the same applied stress (σ). This correlation, plotted in Figure 11, was used in conjunction with the data bounds on Figures 5 and 6 to determine range of life expectancies for various sizes of defects associated with the three hole sizes mentioned above.

8

FAST FRACTURE

FATIGUE CRACK
NOTCH
HOLE
NOTCH
FATIGUE CRACK

FAST FRACTURE
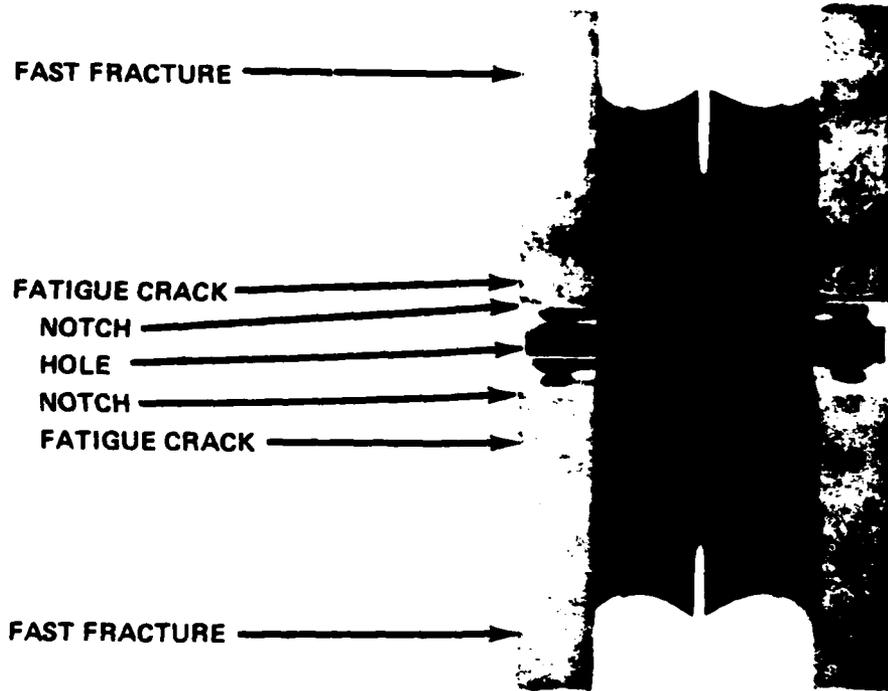
Figure 4. Specimen Fracture Surface
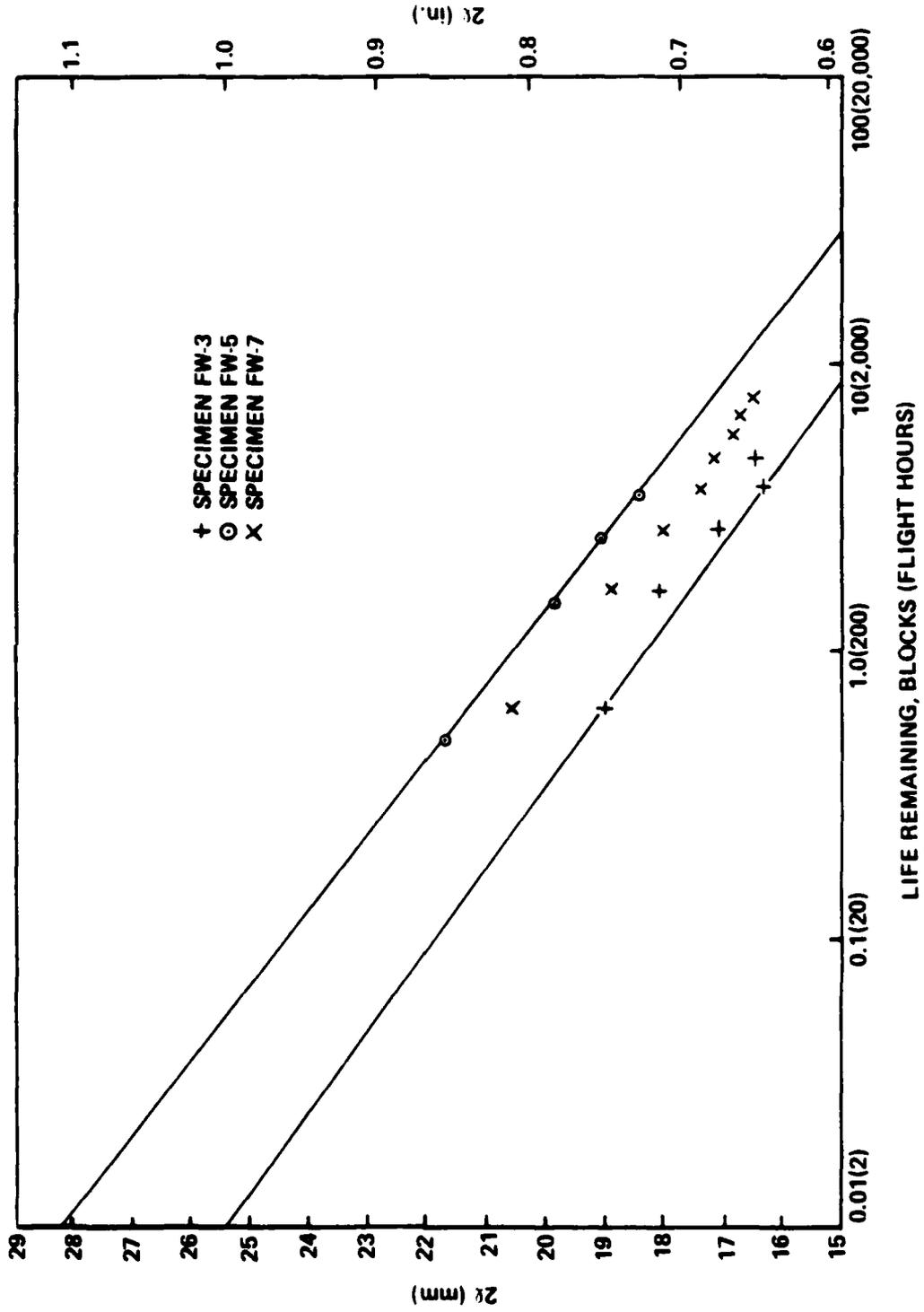
Figure 5. Total Crack Length vs. Fatigue Life Expectancy for Center Cracked
Specimens under 4G Spectrum Loading
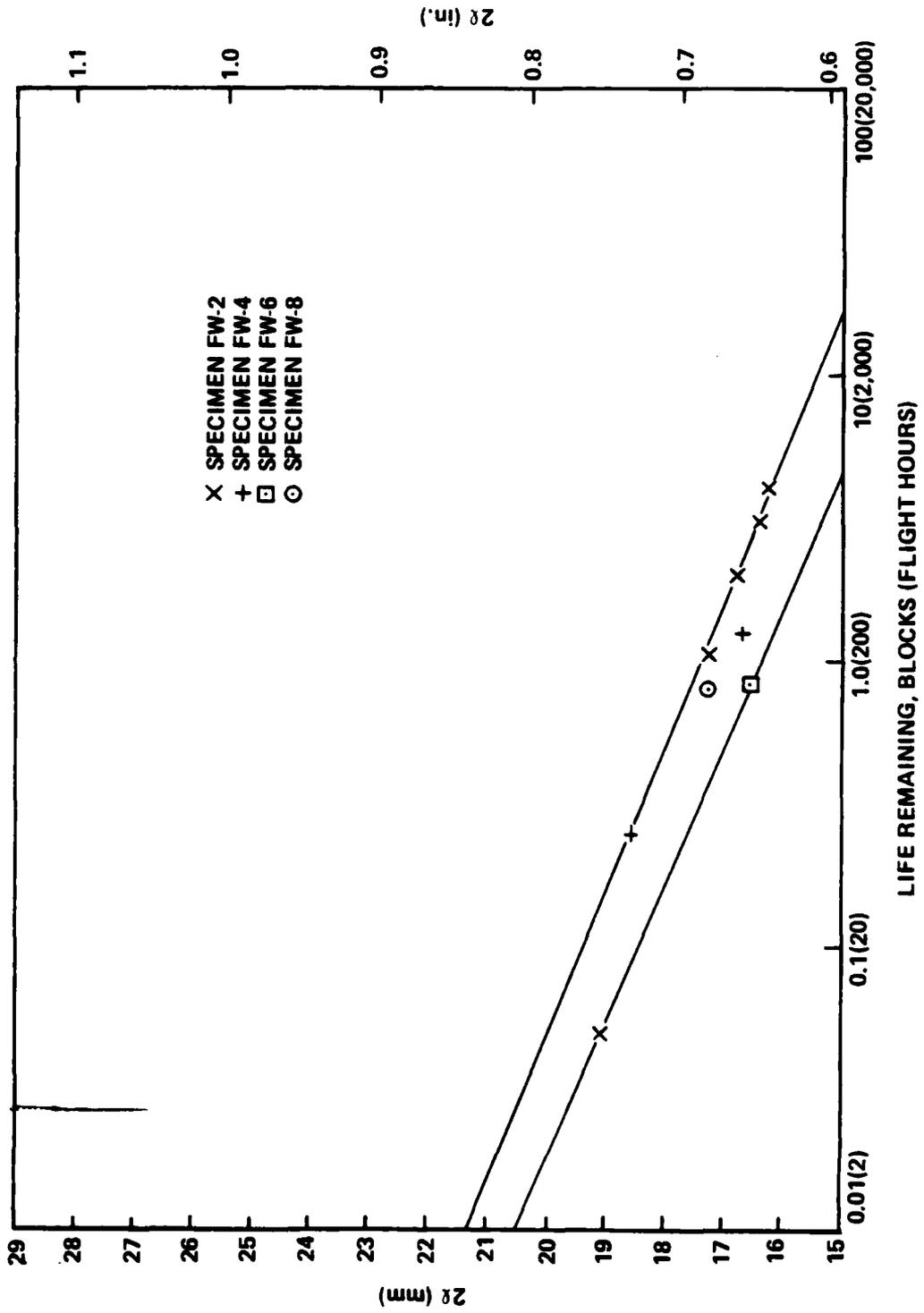
Figure 6. Total Crack Length vs. Fatigue Life Expectancy for Center Cracked
Specimens under 5G Spectrum Loading
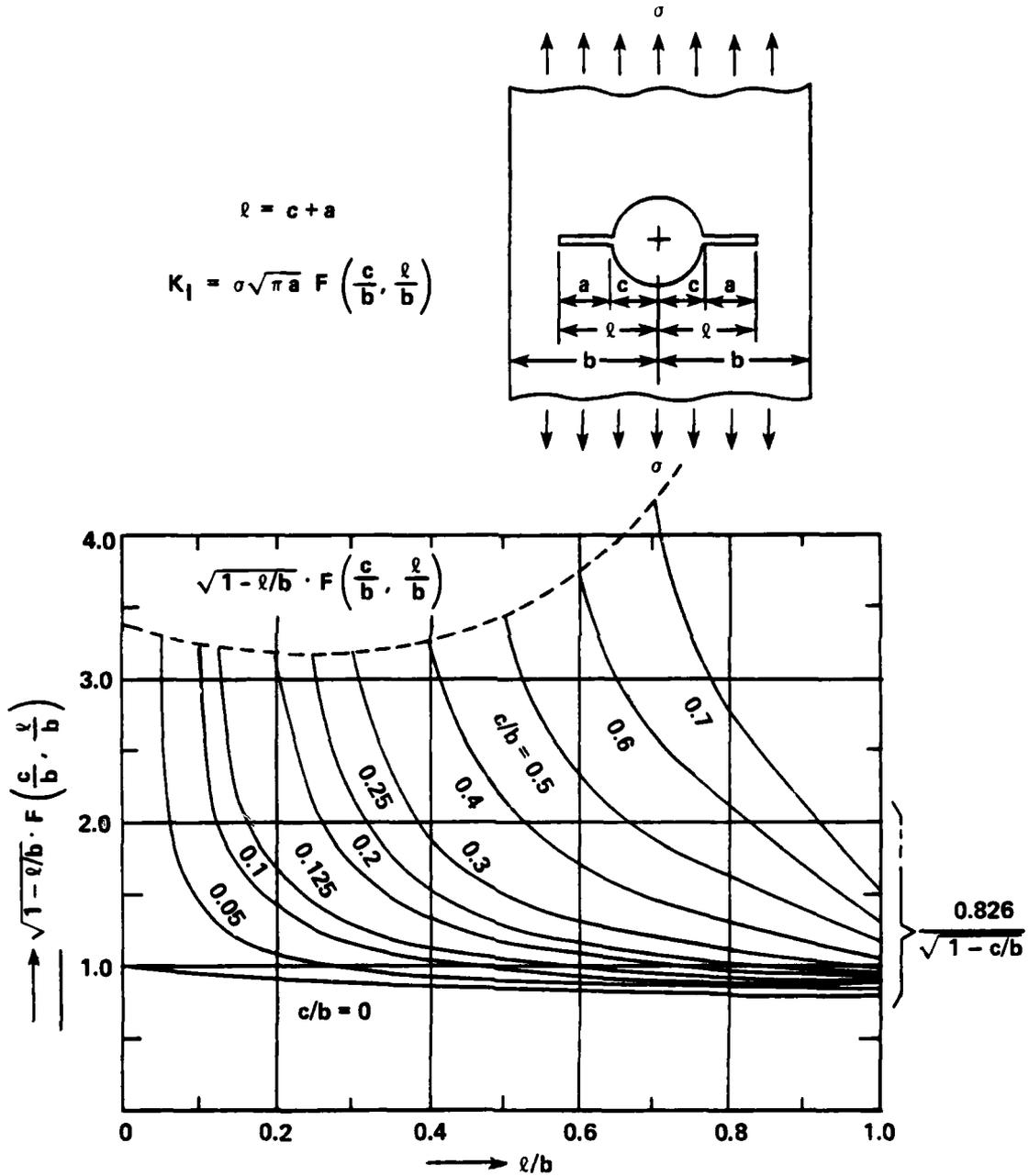
Figure 7. Stress Intensity Solution for Electron Beam Weld
Specimens (reference (c))
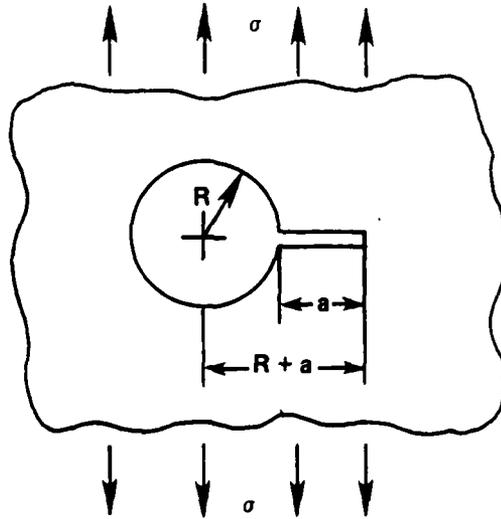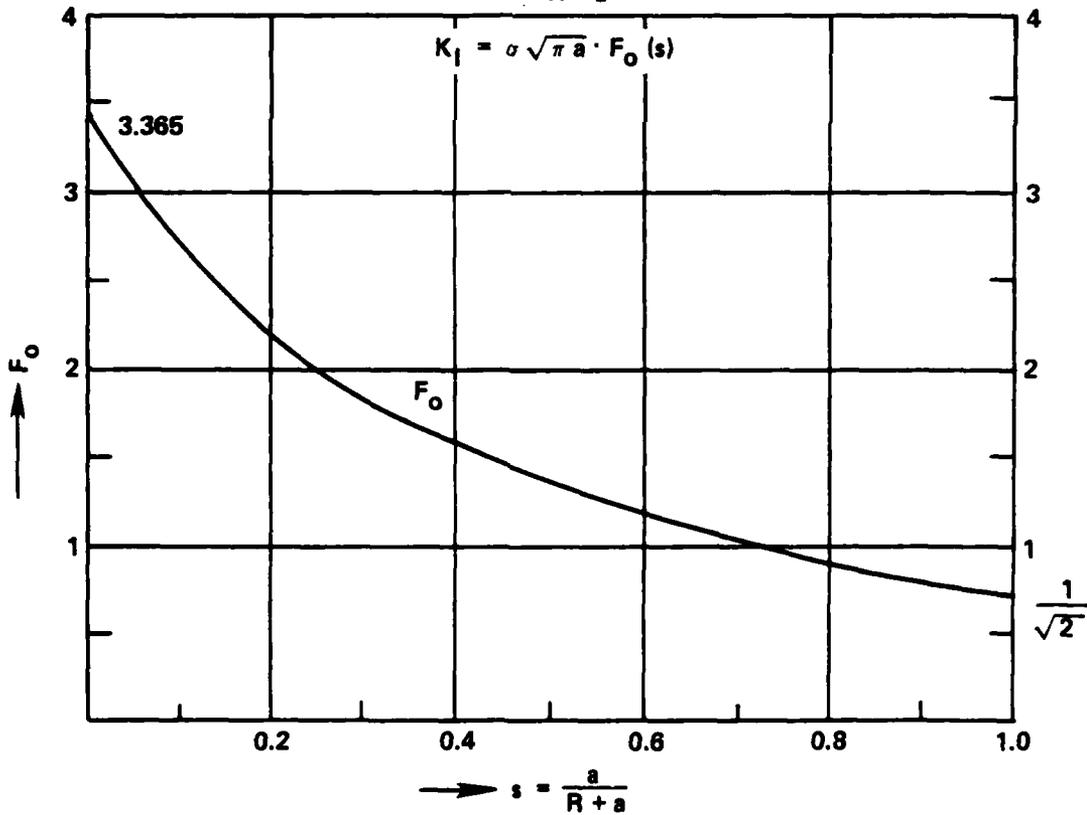
Figure 8. Stress Intensity Solution for Weld Intersect Hole with "Spike" Defect (reference (d))
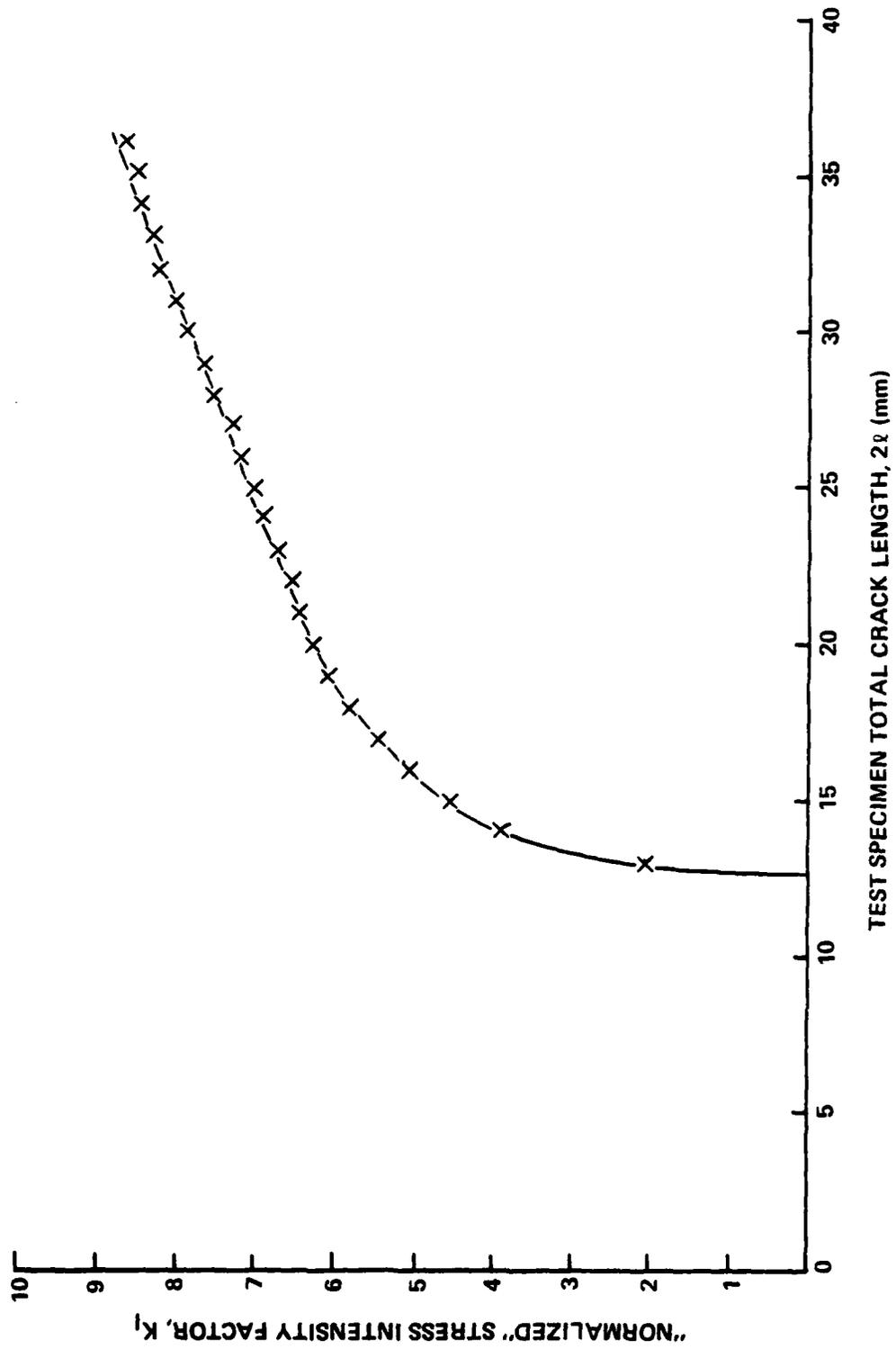
Figure 9. Stress Intensity vs. Crack Length Relationship for Electron Beam Weld Test Specimens
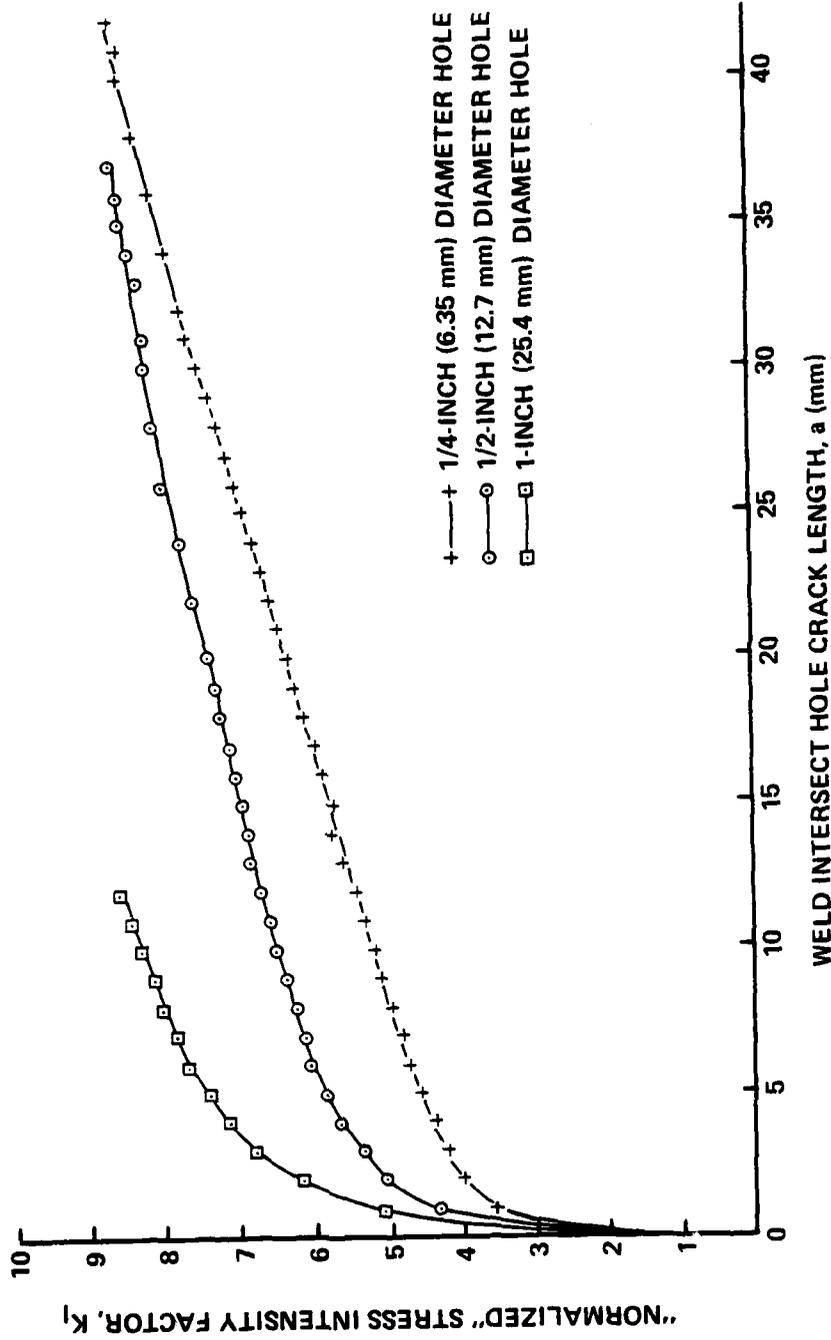
Figure 10. Stress Intensity vs. Crack Length Relationships for Weld Intersect Holes
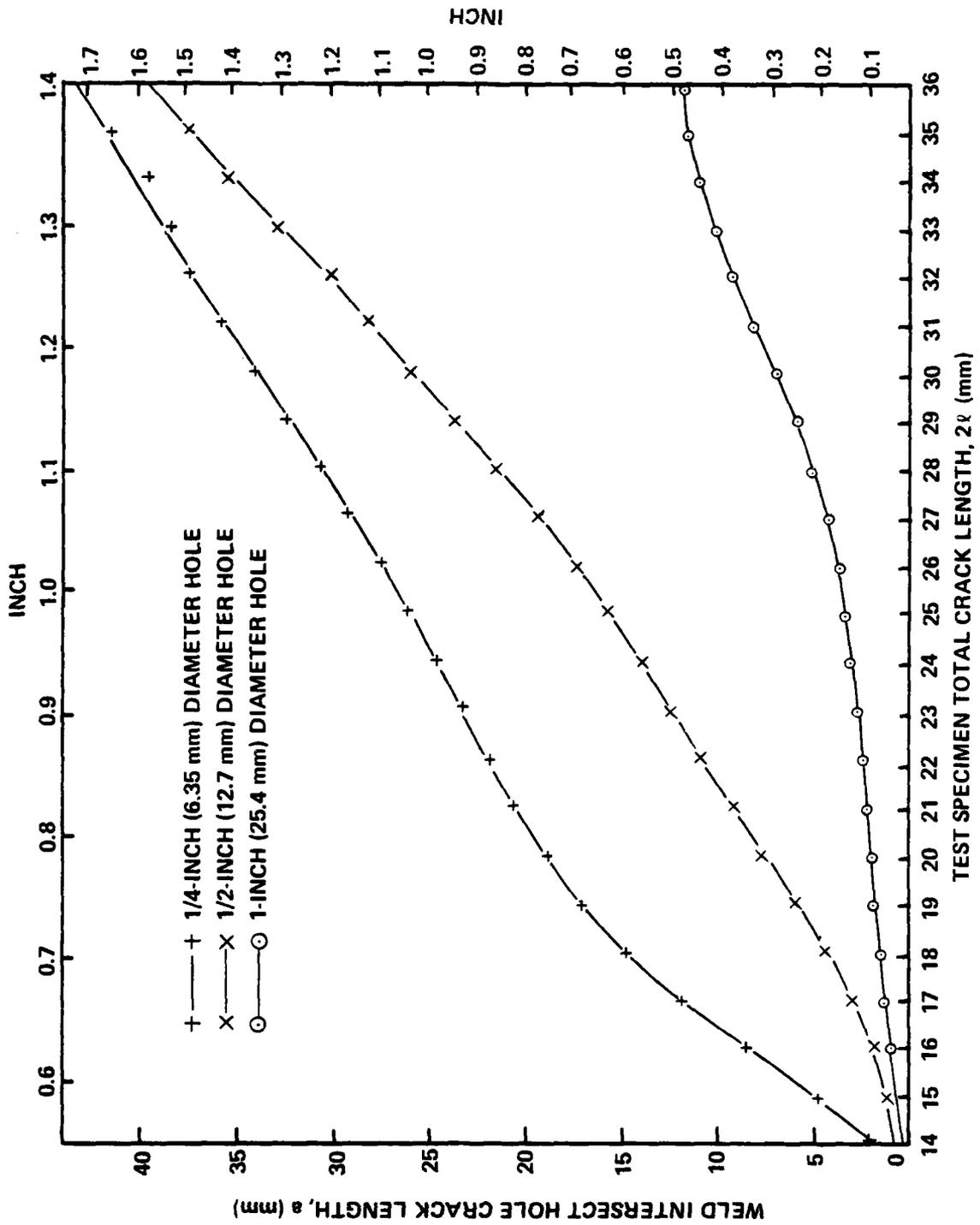
Figure 11. Relationship of Weld Intersect Hole Crack Length to Test Specimen
Total Crack Length for Equivalent Stress Intensity Factors

Table VI summarizes the fatigue life expectancies thus determined. Since the mathematical solutions have been programmed for a Wang 600 calculator, it is also possible to determine life expectancies for other hole sizes and defect sizes.

T A B L E   V I

LIFE EXPECTANCIES FOR F-14 AIRCRAFT WING CARRYTHROUGH SECTIONS FOR VARIOUS SIZE DEFECTS AT WELD INTERSECT HOLES

Hole Size		Defect Size		Life Expectancies					
				4G Loading			5G Loading		
in.	(mm)	in.	(mm)	Flt. Hr.			Flt. Hr.		
0.25	(6.35)	0.039	(1.0)[1]	3940	-	12800	2780	-	11600
0.25	(6.35)	0.079	(2.0)[1]	2940	-	9600	1720	-	7000
0.25	(6.35)	0.118	(3.0)[1]	2240	-	7460	1080	-	4240
0.25	(6.35)	0.158	(4.0)[1]	1980	-	6660	900	-	3480
0.25	(6.35)	0.197	(5.0)[1]	1580	-	5360	610	-	2320
0.25	(6.35)	0.236	(6.0)[1]	1380	-	4700	480	-	1800
0.25	(6.35)	0.276	(7.0)[1]	1210	-	4200	394	-	1440
0.25	(6.35)	0.315	(8.0)[1]	1010	-	3560	286	-	1060
0.25	(6.35)	0.354	(9.0)	830	-	2960	260	-	940
0.25	(6.35)	0.394	(10.0)	690	-	2480	188	-	770
0.25	(6.35)	0.473	(12.0)	470	-	1740	98	-	328
0.25	(6.35)	0.551	(14.0)	300	-	1140	46	-	146
0.25	(6.35)	0.630	(16.0)	180	-	700	19	-	58
0.25	(6.35)	0.709	(18.0)	100	-	410	7	-	20
0.25	(6.35)	0.788	(20.0)	48	-	200		___	
0.25	(6.35)	0.866	(22.0)[1]	20	-	90			
0.50	(12.7)	0.039	(1.0)[1]	2040	-	6900	940	-	3860
0.50	(12.7)	0.079	(2.0)	900	-	3150	290	-	1060
0.50	(12.7)	0.118	(3.0)	500	-	1840	108	-	370
0.50	(12.7)	0.158	(4.0)	300	-	1140	46	-	146
0.50	(12.7)	0.197	(5.0)	198	-	760	22	-	68
0.50	(12.7)	0.236	(6.0)	134	-	540	12	-	34
0.50	(12.7)	0.276	(7.0)	87	-	360	6	-	16
0.50	(12.7)	0.315	(8.0)	59	-	250		___	
0.50	(12.7)	0.354	(9.0)	40	-	174		___	
0.50	(12.7)	0.394	(10.0)	28	-	122		___	
0.50	(12.7)	0.433	(11.0)	19	-	84		___	
1.00	(25.4)	0.039	(1.0)	900	-	3150	290	-	1060
1.00	(25.4)	0.059	(1.5)	280	-	1070	40	-	130
1.00	(25.4)	0.079	(2.0)	90	-	380	6	-	16
1.00	(25.4)	0.098	(2.5)	29	-	128		___	

Note: [1] Calculated by extrapolation of data

D I S C U S S I O N   O F   R E S U L T S

The test and data analysis method described above can be applied in general to problems of fatigue crack propagation under variable amplitude loading (and consequently variable stress intensity amplitude). Its advantage is that it circumvents the uncertain procedure of integrating crack growth (da/dN) vs. stress intensity amplitude (ΔK) curves. Its primary limitation is that several successive graphical interpretations are required, the most uncertain of which is the determination of boundary limits for the crack length vs. life expectancy data.

In this test program there were at least two other factors also affecting the applicability of the results to aircraft. The first is the question of how closely the 4G and 5G load spectra duplicate the actual loading of the airframe in service. The second is the fact that fatigue cracks were substituted for actual weld flaws. Fatigue cracks undoubtedly have smaller tip radii than do spike defects. Therefore, in service, fatigue lives should be longer than those indicated. This is because of the added time required to initiate a fatigue crack at the tip of the defect. Such a time delay could be estimated from fatigue crack initiation data.

The practical significance of this test program is contained in the predicted life times shown in Table VI. At present the Scheduled Depot Level Maintenance (SDLM) interval for the F-14 aircraft is 30 months, which corresponds to 500 to 700 flight hours (reference (e)), and during maintenance the aircraft are not inspected specifically for defects or cracks in the wing carrythrough section. Clearly no defect size listed in Table VI that falls within the range of the data is tolerable under 5G loading, and most are not tolerable under 4G loading. Most of the lifetimes for small defects determined by extrapolation of the test data are tolerable, but these are subject to larger errors than those that fall within the range of the data. Thus, one must conclude that on the basis of the available data, an F-14 wing carrythrough section cannot tolerate defects at weld intersect holes as large as 0.079 inch (2.0 mm) for 1/2 inch (12.7 mm) diameter holes, 0.354 inch (9.0 mm) for 1/4 inch (6.35 mm) diameter holes, and 0.039 inch (1.0 mm) for 1 inch (25.4 mm) diameter holes.

ACKNOWLEDGEMENT

The author wishes to acknowledge the assistance of T. Di Fore, J. Bethke and
E. Tankins in preparing the spectrum loading programs for the test equipment,
and of E. Fogarty in conducting the flaw propagation tests.

R E F E R E N C E S

(a) "Spectra for Evaluation of Flaws at EB-Weld Intersect Holes," (GAC Request for Test F14-4077), Grumman Aerospace Corporation, 12 Oct 1971.

(b) Ketcham, S. J., Neu, C. E., and Goldberg, S., "Stress Corrosion Testing of Titanium Alloys," AGARD Conference Proceedings No. 98, Oct 1971.

(c) The Stress Analysis of Cracks Handbook, Del Research Corporation, 1973, p 19.9.

(d) The Stress Analysis of Cracks Handbook, p 19.2

(e) J. Hoffner, Naval Air Systems Command Headquarters, Washington, DC, Private Communication, Nov 1977.